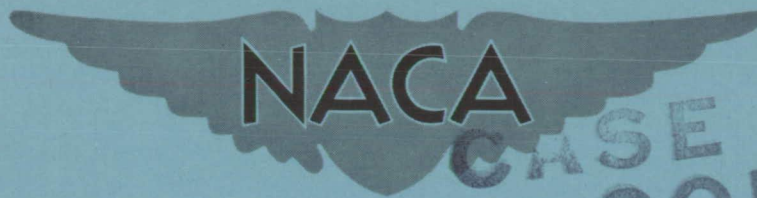355

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RESEARCH MEMORANDUM

SOME DESIGN CONSIDERATIONS PERTINENT TO THE ROUGH-AIR

BEHAVIOR OF AIRPLANES AT LOW ALTITUDE

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RESEARCH MEMORANDUM

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SUMMARY

      For certain types of military operations that will require flights
of 1 or 2 hours duration at altitudes of 1,000 feet or less at high sub-
sonic speeds, study has indicated that turbulence is of importance in
regard to the precision of flight and the efficiency of the crew. Anal-
ysis indicated that a reduction of disturbed motions to one-third the
current values may be required.

      Examination of design variables that might lead to the desired
acceleration level showed the major factors to be increased wing loading
or reduced lift-curve slope by means of sweep, reduced aspect ratio, or
increased flexibility. Changes in stability for airplanes with satis-
factory characteristics were not significant but, for configurations
involving swept wings or low damping of airplane motions, the adverse
effects produced can be significantly reduced by artificial damping.

INTRODUCTION

      There are certain types of military operations that will require
flights of 1 or 2 hours duration at altitudes of 1,000 feet or less at
high subsonic speeds, speeds that must be maintained. Rough air is
encountered about 30 to 40 percent of the time at low altitudes, and
the National Advisory Committee for Aeronautics in a study found that
turbulence is of importance not only for structural strength but also
in regard to the precision of flight and also in regard to the crew's
efficiency and well-being. The study indicated that even moderately
rough air would be troublesome and the problems of flight precision and
crew reactions can be considered as new problems that may modify the
design of the airplane.

      The purpose of this discussion is to examine some of the design vari-
ables that have a bearing on the problem. After treating the question of

what happens to an airplane and the conditions that appear desirable, the direct effects of turbulence are considered and are followed by a discussion of the influence of some variables affecting the motions of the airplane.

SYMBOLS

W	weight, lb
V	forward speed, fps
U	gust velocity, fps
$dC_L/d\alpha$	slope of lift curve, per radian
Λ	sweep angle, deg
S	wing area, sq ft
ρ	air density, slugs/cu ft
Δn	acceleration increment, g
$\bar{c}$	wing mean aerodynamic chord, ft
c'	wing mean geometric chord, ft
A	aspect ratio
$\Delta\theta$	pitch-angle increment
μ_g	mass ratio, $2W \left/ \rho g \dfrac{dC_L}{d\alpha} Sc'\right.$
f	frequency, cps
Ω	frequency, radians/ft
Φ_i	power density, $(ft/sec)^2$ ft
L	scale of turbulence, ft
$T_{1/2}$	time to damp to half-amplitude, sec
$C_{1/10}$	cycles to damp to 1/10 amplitude

p   period, sec

dC_m/dC_L slope of pitching-moment curve

RESULTS AND DISCUSSION

Airplane reactions of concern as to the precision of flight and crew's reactions are illustrated in figure 1 by these short samples of acceleration and direction records from flight at a Mach number of only 0.60 at an altitude of 1,500 feet. Although this flight is in moderate to severe rough air, the acceleration record indicates that the crew was being continually jolted. The heading record is of interest in that while the crew was being jolted, the airplane developed a yawing oscillation with a period of about 3 seconds with an amplitude of ±1.3°, an oscillation not present in smooth air. This amplitude would correspond to a miss distance of 22 feet at a 1,000-foot range or 22 mils. Similar behavior has been noted for other airplanes, as for example the X-5 with the wings swept 59°.

The feelings of the crew will depend on both the acceleration intensity and the frequency. Figure 2 is another example of the jolting that can be experienced. This acceleration record is a 12-second section from a 2-minute flight (21 miles) at 1,500 feet with an F-86 airplane flying at $M \approx 0.85$. The complete run shows accelerations up to 1.5g while this particular section shows small rapid oscillations at about 3 or 4 per second with peaks of about 0.2g. Although the pilot had made many runs at Mach numbers of 0.4 and 0.8, he noted that this run of 2 minutes was about his limit. He felt that, at this speed and altitude, the degree of concentration and effort required to control the airplane was so great as to jeopardize the safety of the airplane if such flights were continued.

The decision as to how much the reactions of the airplane should be reduced is a difficult one since it is subjective and little factual information is available. From the short runs made with the F-86 airplane, it appears that moderate rough air results in a ride that is near the safety limit. For flights of 1 or 2 hours in rough air, a substantial reduction in the acceleration level would appear to be required; perhaps, a reduction to about 30 percent of the level shown. McFarland in his book on human factors in air transport design (see ref. 1) indicates that the work of Reihe and Meister showed that, although accelerations of ±0.2g at 3 to 5 cps would be dangerous, a level of 0.04g at these frequencies would be merely disagreeable. This is a reduction to 20 percent of the level shown on the F-86 records and assumes that a disagreeable ride would be tolerable. In regard to airplane motions, a deviation of 5 mils is considered satisfactory; thus, the motions of the B-45 airplane at 22 mils would have to be decreased to about 25 percent of

that value to be useful under the conditions specified for this mission. Since all considerations indicate a reduction to about 20 or 30 percent, the acceleration record for the F-86 airplane will be used as a standard, and a reduction by a factor 3 in acceleration or to 30 percent will be the criterion of satisfactory behavior for moderately rough air used herein.

The factors that are pertinent for obtaining the desired reduction are indicated in the following relationship:

$$\Delta n \approx \frac{\rho UV}{2} \frac{dC_L/d\alpha}{W/S} \pm \frac{\rho V^2}{2} \frac{dC_L/d\alpha}{W/S} \Delta\theta \qquad (1)$$

The acceleration can be considered as composed of two effects: the direct effect of the gust consisting of such quantities as airspeed, air density, lift-curve slope, and wing loading which determine the magnitude of the disturbance of the airplane and the indirect effects due to the resulting airplane motions which are represented by the pitch angle $\Delta\theta$. Since the speed, air density, and gust velocity are specified in this problem, the major elements at the disposal of the designer are changes in lift-curve slope, wing loading, and the disturbed airplane motions.

What can be accomplished by working on the lift-curve slope and wing loading will be taken up first. The remainder of the discussion will be concerned with the second term representing the influence of airplane motion. The indirect effects of various factors such as, sweep, static margin, lack of a tail surface, and artificial damping will be touched on.

Direct Effects

Three methods of obtaining a low lift-curve slope are to reduce the aspect ratio, sweep the wing, or introduce flexibility particularly for a sweptback wing. The effect of aspect ratio and sweep are used in combination many times. Gust-tunnel investigations, references 2 and 3, have indicated the effectiveness of both quantities and figure 3 shows that these investigations are borne out by flight experience. Figure 3 indicates that, if an unswept-wing airplane encountered a given value of acceleration, then the ordinate represents the acceleration that a swept-wing airplane would experience in the same rough air with equal frequency. In order to obtain the line of circle test points labeled 35°, an F-80 and an F-86 airplane were flown side by side in the same rough air at a Mach number of about 0.6. In this case, the F-80 with about the same wing loading and aspect ratio as the F-86 was used as the reference, and the acceleration increments for equal frequency of occurrence were plotted. The square test points labeled 59° were obtained from go-and-return flights

of the X-5 airplane in rough air at a Mach number of about 0.7. Alternate runs with the wings swept 20° and 59° were made. The X-5 data for 20° sweep corrected slightly to 0° were the reference conditions to obtain the square test points. The solid lines represent the relation between accelerations for the airplanes if the lift-curve slope is assumed to be the only factor. As can be noted, the agreement is fairly good.

Figure 3 indicates that reducing the lift-curve slope through the use of sweep is quite effective. Since the F-86 acceleration record is the basis for the reduction desired, it can be seen that, if the wings were swept to 60° by rotation, the acceleration could be reduced from say 0.6g to about 0.4g, 30 percent or about half the desired reduction. If this same 30-percent reduction were to be obtained by reducing aspect ratio, the aspect ratio would have to be decreased say from 6.0 to 2.0. Unless extreme values of sweep or aspect ratio are utilized, it does not appear practical to obtain the reduction to 30 percent in this way. As a matter of fact, the relation given indicates that for aspect ratios below about 1.5 the effect of sweep is not significant.

Flexibility utilizing washout under load is the third way of obtaining low lift-curve slopes. About the only experimental evidence is contained in reference 4. This gust-tunnel investigation of a 45° sweptback wing, hinged at the root, showed that a 21-percent reduction in acceleration could be obtained if the airplane did not pitch. The wing deflection corresponded to 20 inches per g at the tip of a wing with a span of 100 feet. Unfortunately, these tests also indicated that adverse pitch due to forward movement of the aerodynamic center as the wings deflected canceled half the gain. It is apparent, therefore, that, although the introduction of flexibility can be beneficial, the net gain may depend on the induced airplane motions to a high degree and these motions may require careful consideration. So far no full-scale experimental results are available to assess this phase of the problem.

The effect of wing loading on accelerations due to a gust are well-known, reference 2, but for convenience are shown in figure 4 for wings of various sweep. Simple calculations for a single gust encounter have been made for a 20-fps gust (about the maximum experienced by the F-86) and a flight speed of 1,000 fps. Values of the acceleration increment for wing loadings from 50 to 300 lb/sq ft are shown for sweeps of 0°, 50°, and 60°. A delta wing of 60° would follow the 60° line quite closely. The curves indicate that, for the wings shown, the wing loading for an unswept-wing airplane would have to be increased from 50 to about 200 lb/sq ft to reduce the acceleration to 30 percent. If a shift is made from an unswept wing to one swept 60°, the wing loading would have to be increased to only 100 lb/sq ft to achieve the reduction.

Indirect Effects

What can be done by modifying the second term of equation (1), which represents in principle the effect of airplane motion, is not so obvious. The subject of airplane motions is complex but the magnitude of changes in acceleration have been studied since the equation indicates this term may increase or decrease the acceleration. Since continuous rough air is being dealt with, generalized harmonic analysis for random disturbances, references 5 and 6, has been utilized in the subsequent studies.

The remainder of the discussion deals with possible benefits of modifying the stability of stable well-damped airplanes, the influence of adverse moments due to sweep on the benefits just indicated, comparison of tailed and tailless configurations, and the use of artificial damping for poorly damped motions. Although it was found that most factors were not significant in the problem under consideration, the study of configuration and artificial damping indicated that some adverse effects could be eliminated or reduced.

Input spectrum.- For the analyses of the effects of modifying the stability of well-damped airplanes and the effect of adverse moments due to sweep, the input spectrum shown in figure 5 was used. The curve of power density Φ_i as a function of frequency is based primarily on airspeed fluctuation data obtained on an L-5 airplane operating in moderately rough air at an altitude of 400 feet. These data which are in agreement with other samples were used to arrive at a fitted curve for isotropic turbulence. The curve shown in figure 5 is for a scale length L of 300 feet.

For the study of tailed and tailless configurations and artificial damping, spectra were for a lower level of turbulence. Since the basic investigation involved experimental studies with rocket models, it was more convenient to utilize the associated analyses. Since in this study relative effects on models are being assessed, the actual intensity is not significant.

Unswept-wing airplanes.- Figure 6 and table I give the characteristics of the airplane family used to study changes in dynamic stability obtained by varying the moment-curve slope dC_m/dC_L and the mass ratio μ_g. The moment-curve slope dC_m/dC_L was varied from -0.03 to -0.08 and the mass ratio was varied from 10 to 200. The airplanes had a weight of 100,000 pounds and were geometrically similar with a flight speed of 1,118 fps. Figure 6 indicates that the heavily loaded airplane had a short period from 5.6 to 3.1 seconds, damping to half-amplitude in 0.9 seconds, whereas the lightly loaded airplane has an infinite short period and is heavily damped. Some of these airplanes represent extreme variations.

The response transforms were obtained by computing the step function according to methods of reference 7 and then transforming the results to the frequency plane. Although flight at high Mach number is assumed, two-dimensional incompressible unsteady-lift functions and low-speed lift-curve slopes were used. This was done since the available evidence, reference 8 and other studies, is very inconclusive as to the proper functions for gust calculations at high Mach number. Also, the airplanes were assumed to be rigid; therefore, structural vibrations are not included in the response calculations.

The response transforms were then multiplied by the input spectrum of figure 5 to obtain output spectra. As noted in reference 5, the area under the output spectrum is the mean square acceleration and the square root of this quantity, the root-mean-square acceleration increment, will be used as a measure of the airplane behavior.

The results are given in figure 7 as a function of wing loading with dC_m/dC_L as the parametric variable. For comparison, the root-mean-square acceleration increment for the F-86 record of which figure 2 is a portion was 0.21 so the criterion of reduction used herein would imply that a satisfactory value would be about 0.07. As previously noted the direct effect of wing loading seems to be the dominant factor and the influence of stability is small and unimportant for the problem at hand. It might be noted that at low wing loadings the effect of stability was negligible, whereas at high wing loadings a reduction in stability tends to reduce the accelerations slightly. Other studies have shown similar results; but, it has been found that, depending on the mass ratio and geometry, the effect of increased stability is sometimes favorable as indicated in reference 9.

Swept-wing airplanes.- Since the root section of a swept wing penetrates a gust before the wing tips, an adverse pitching moment is produced similar to that mentioned earlier for the flexible sweptback wing. An analysis was made, therefore, to see whether this factor might cancel the direct benefits obtained through reducing the lift-curve slope. The airplane characteristics are given in figure 8 and table II. As in the previous case the weight was kept at 100,000 pounds, the mass ratio was varied from 10 to 200, and the sweep was varied from 0° to 60° with the aspect ratio kept unchanged. Strip theory was used to modify the unsteady-lift functions to account for sweep. The speed was the same as for the unswept-wing family. The airplanes were assumed to be neutrally stable (aperiodic), and, as indicated, the time to damp to half-amplitude varied from 0.12 second to about 0.90 second as the wing loading increased.

In figure 9, the root-mean-square acceleration increment is again shown as a function of wing loading. The lowest curve, indicated by the diamond symbols, indicates the direct effect of lift-curve slope. This

curve was obtained by multiplying the root-mean-square acceleration for 0° sweep by the ratio of lift-curve slopes for the 60° and 0° sweep cases. The difference between this curve and the 60° curve with triangle symbols is used as a measure of the influence of pitch. At low wing loadings, the three curves coincide indicating that the pitch had effectively canceled the effect of reduced lift-curve slope. As the wing loading increases, the influence of pitch is decreased so that most of the benefit of sweep is realized. At a wing loading of 160 lb/sq ft, the adverse pitch increases the acceleration 14 percent. Such adverse effects indicate increased angular motions of the airplane that might require modification for gunnery or bombing considerations.

Airplane configuration and artificial damping.- So far all configurations had well-damped motions; but, of considerable concern, are low-damped airplane motions such as illustrated by the heading record shown in figure 1. A configuration that may well have low damping in pitch is one without a horizontal tail. For such configurations, the motions of the airplanes are significant in their own right and can also lead to increased accelerations that cancel any benefits of increased wing loading or reduced lift-curve slope as for the cases just discussed.

The NACA has initiated exploratory studies of the problem by means of rocket models and analysis. The results that bear on the specific mission will now be discussed preceded by a brief comparison of experiment and calculation to provide a measure of the validity of this and the preceding analyses.

Figure 10 gives the configurations tested and their dynamic characteristics. The models consisted of the same forebody and wing but, as indicated by the dashed lines, in one case the body was extended and a horizontal tail added. The inset figures for period and cycles to damp to 1/10 amplitude indicate that both models were very stable but the damping was poor when compared with the criterion of one cycle to damp to 1/10 amplitude.

The output spectra with the corresponding calculated curves are shown in figure 11 for $M \approx 0.81$ whereas the corresponding values of root-mean-square acceleration increment are given in figure 12. The discrepancy between calculation and experiment represents the combined effects of errors in calculating the motion and accounting for the temporal character of the input spectra during a rocket flight. The spectra shown represent about the greatest discrepancy found and the agreement is considered fairly good. Figure 12 gives an over-all picture of the results and indicates that, when all the experimental samples are considered, the tailless model gave good results (owing in part to the fact that stability derivatives were available from tests of similar models) and the calculated values were low for the tailed model.

For the purpose of this paper, the experiments were not used directly but analyses with the two configurations adjusted to give the same natural frequency were made. In addition to the calculation for the configurations, the effect of adding rate damping to both models was also analyzed. The damping systems assumed control deflections proportional to angular velocity and the black areas on the models (fig. 13(b)) indicate that a trailing-edge flap was used for the tailless model and an all-moving tail for the tailed model. For the tailless model the flap had a chord of 0.25c and the system would respond to 14 cps. The frequency response of the control system for the tailed model was the same.

The calculated output spectra, in figure 13, show by comparing the solid curves that adding a tail reduced the root-mean-square acceleration increment 20 percent. Although this gain is significant it is not as great as might be expected and does not rule out tailless configurations. Comparison of the dashed and solid curves, figure 13, indicates that the use of artificial damping leads to very significant gains. With damping, the output spectra are very flat and the root-mean-square acceleration increments were reduced about 50 percent.

The effect of added damping is shown in a different form in figure 14 where the calculated frequency distributions are plotted for the four cases as the number of peaks per mile as a function of the acceleration increment. Inspection of figure 14 indicates that the added damping has increased the number of small acceleration peaks for both models by increasing the response somewhat at high frequencies. At high acceleration, the benefit of the tail and of the use of artificial damping show up quite clearly, with the damped tailed model having the lowest acceleration at 10 per mile and the damped tailless model next. It is obvious from these results that, for low-damped motions, the use of artificial damping can reduce the accelerations due to airplane motion so that the direct benefits can be utilized if high wing loading and reduced lift-curve slope lead to such motions. It might be noted that the large gains are made only for poorly damped configurations as tests of a rate auto-pilot in a subsonic stable airplane indicated a reduction of only 7 percent, reference 10.

CONCLUDING REMARKS

In conclusion the discussion has indicated that if a reduction in airplane response in rough air by a factor of 3 is required for high-subsonic-speed low-altitude flight:

(a) Increasing the wing loading and reducing the lift-curve slope through sweep, reduced aspect ratio, or increased flexibility will be the major factors.

(b) The effect of moderate changes in stability for airplanes with satisfactory characteristics does not appear significant. For configurations using swept wings and those involving low damping of airplane motions, adverse angular displacements may cancel the benefits of other changes but the adverse effects can be significantly reduced by artificial damping.

It might be noted in closing that solution of the gust problem for this mission may introduce other serious problems in regard to handling qualities or increased landing and take-off speeds.

Langley Aeronautical Laboratory,
    National Advisory Committee for Aeronautics,
        Langley Field, Va., September 17, 1953.

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TABLE I

CHARACTERISTICS OF UNSWEPT WING AIRPLANES

[Gross weight, 100,000 lb]

	$\frac{dC_m}{dC_L} = -0.03$			$\frac{dC_m}{dC_L} = -0.05$			$\frac{dC_m}{dC_L} = -0.08$		
Mass ratio	10	80	200	10	80	200	10	80	200
Pitching moment of inertia slug-ft²	2,889,000	718,500	395,200	2,889,000	718,500	395,200	2,889,000	718,500	395,200
Air density, slugs/cu ft . . .	0.00238	0.00238	0.00238	0.00238	0.00238	0.00238	0.00238	0.00238	0.00238
Forward velocity, fps . . .	1,118	1,118	1,118	1,118	1,118	1,118	1,118	1,118	1,118
Wing aspect ratio . . .	3.1	3.1	3.65	3.1	3.1	3.1	3.1	3.1	3.1
Wing lift-curve slope, per radian . . .	3.65	3.65	3.65	3.65	3.65	3.65	3.65	3.65	3.65
Wing mean aerodynamic chord, ft . . .	30.5	15.2	11.3	30.5	15.2	11.3	30.5	15.2	11.3
Gross wing area, sq ft . . .	2,512	625	343	2,512	625	343	2,512	625	343
Wing loading, lb/sq ft . . .	39.8	160.0	291.5	39.8	160.0	291.5	39.8	160.0	291.5
Horizontal distance from center of gravity of airplane to aerodynamic center of wing, ft . . .	5.02	2.50	1.86	4.44	2.21	1.64	3.54	1.76	1.31
Tail lift-curve slope, per radian . . .	3.57	3.57	3.57	3.57	3.57	3.57	3.57	3.57	3.57
Tail mean aerodynamic chord, ft . . .	12.5	6.23	4.62	12.5	6.23	4.62	12.5	6.23	4.62
Gross tail area, sq ft . . .	469	117	64	469	117	64	469	117	64
Horizontal distance from center of gravity of airplane to aerodynamic center of tail, ft . . .	84.3	42.0	31.2	85.0	42.4	31.4	86.0	42.8	31.8
Distance between trailing edge of wing mean aerodynamic chord to leading edge of tail mean aerodynamic chord, ft . . .	54.1	27.0	20.0	54.1	27.0	20.0	54.1	27.0	20.0
Tail efficiency factor	0.8	0.8	0.8	0.8	0.8	0.8	0.8	0.8	0.8
Downwash factor	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5

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TABLE II

CHARACTERISTICS OF SWEPT WING AIRPLANES

$$\left[\text{Gross weight, 100,000 lb; } dC_m/dC_L = 0\right]$$

	$\Lambda = 0°$			$\Lambda = 45°$			$\Lambda = 60°$		
Mass ratio	10	80	200	10	80	200	10	80	200
Pitching moment of inertia slug-ft²	2,594,000	653,000	355,600	2,814,000	708,200	382,700	3,261,000	815,100	439,800
Air density, slugs/cu ft . .	0.00238	0.00238	0.00238	0.00238	0.00238	0.00238	0.00238	0.00238	0.00238
Forward velocity, fps . .	1,118	1,118	1,118	1,118	1,118	1,118	1,118	1,118	1,118
Wing aspect ratio	3.0	3.0	3.0	3.0	3.0	3.0	3.0	3.0	3.0
Wing lift-curve slope, per radian	3.6	3.6	3.6	3.18	3.18	3.18	2.56	2.56	2.56
Wing mean aerodynamic chord, ft	28.9	14.5	10.7	30.1	15.1	11.1	32.4	16.2	11.9
Gross wing area, sq ft . . .	2,507	627	340	2,724	681	370	3,147	787	427
Wing loading, lb/sq ft . . .	39.9	159.5	294.1	36.7	146.8	270.3	31.8	127.1	234.2
Horizontal distance from center of gravity of airplane to aerodynamic center of wing, ft	6.42	3.21	2.37	6.70	3.35	2.47	7.20	3.60	2.65
Tail lift-curve slope, per radian	3.6	3.6	3.6	3.18	3.18	3.18	2.56	2.56	2.56
Tail mean aerodynamic chord, ft	12.9	6.46	4.76	13.5	6.74	4.96	14.5	7.24	5.34
Gross tail area, sq ft . . .	501	125	68.1	544	136	73.9	629	157	85.4
Horizontal distance from center of gravity of airplane to aerodynamic center of tail, ft	80.3	40.2	29.6	83.7	41.8	30.8	90.0	45.0	33.1
Distance between trailing edge of wing mean aerodynamic chord to leading edge of tail mean aerodynamic chord, ft	61.8	30.9	22.8	76.9	38.5	28.3	92.5	46.3	34.1
Tail efficiency factor	0.8	0.8	0.8	0.8	0.8	0.8	0.8	0.8	0.8
Downwash factor	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5

TIME HISTORIES FOR B-45 AIRPLANE IN ROUGH AIR
M≈0.60; ALTITUDE, 1,500 FT
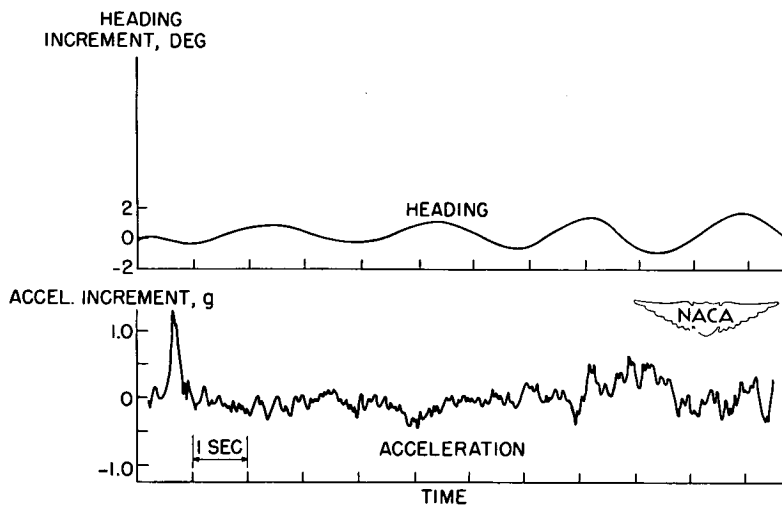

Figure 1

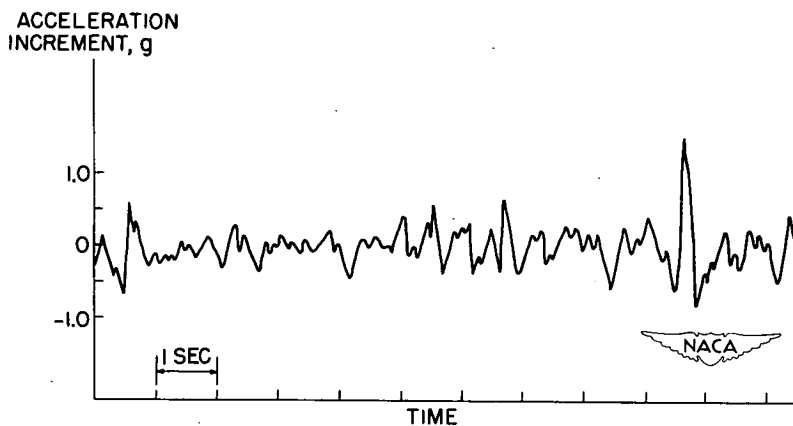ACCELERATION RECORD FOR F-86 AIRPLANE
IN ROUGH AIR
M≈0.85; ALTITUDE, 1,500 FT

Figure 2

FLIGHT TEST RESULTS ON SWEPT-WING AIRPLANES

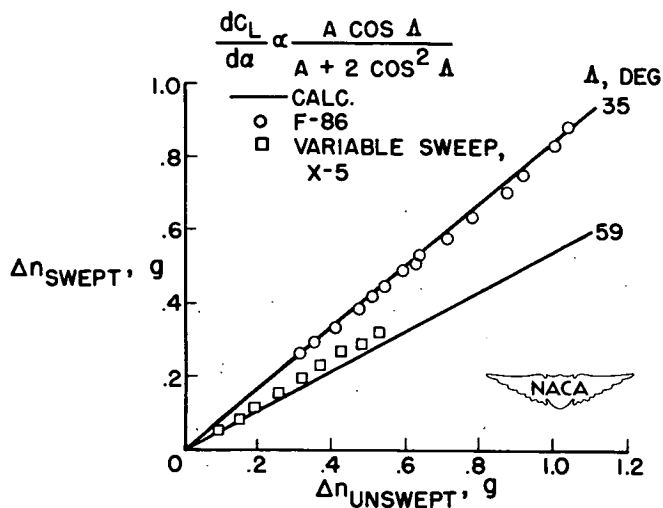$$\frac{dc_L}{d\alpha} \propto \frac{A \cos \Lambda}{A + 2 \cos^2 \Lambda}$$

——— CALC.

o   F-86

□   VARIABLE SWEEP, X-5

Δn_{SWEPT}, g

$\Delta n_{\text{UNSWEPT}}$, g

Λ, DEG

35

59

Figure 3

EFFECT OF SWEEP AND WING LOADING
ON GUST ACCELERATION

U = 20 FPS; ALTITUDE, 1,000 FT; V = 1,000 FPS

ACCELERATION
INCREMENT, g

Λ, DEG

0

50

60

WING LOADING, LB/SQ FT

Figure 4

SPECTRUM OF TURBULENCE USED

POWER DENSITY, Φ_i, $\left(\dfrac{FT}{SEC}\right)^2$ FT

$$\Phi_i(\Omega) = \Phi_i(0)\ \frac{1+3L^2\Omega^2}{(1+L^2\Omega^2)^2}$$

$$L = 300\ FT$$

$$\Phi_i(0) = 783\left(\frac{FT}{SEC}\right)^2 FT$$
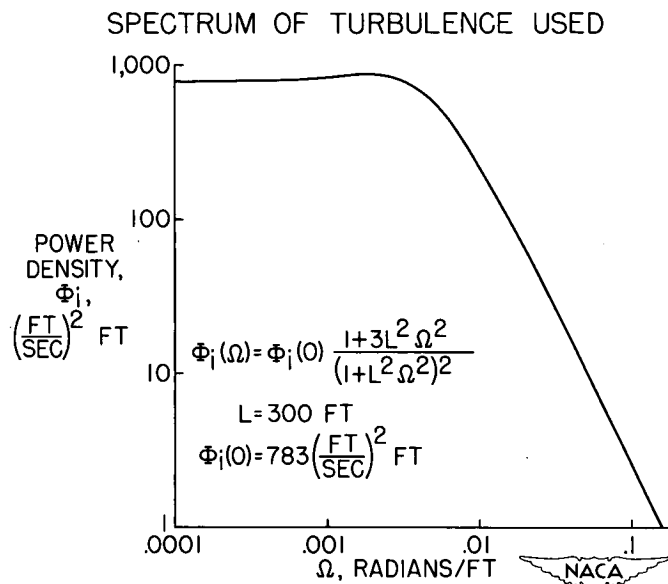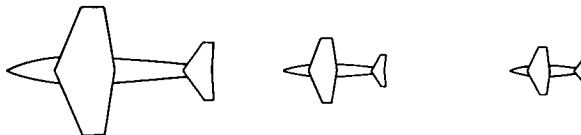
Ω, RADIANS/FT
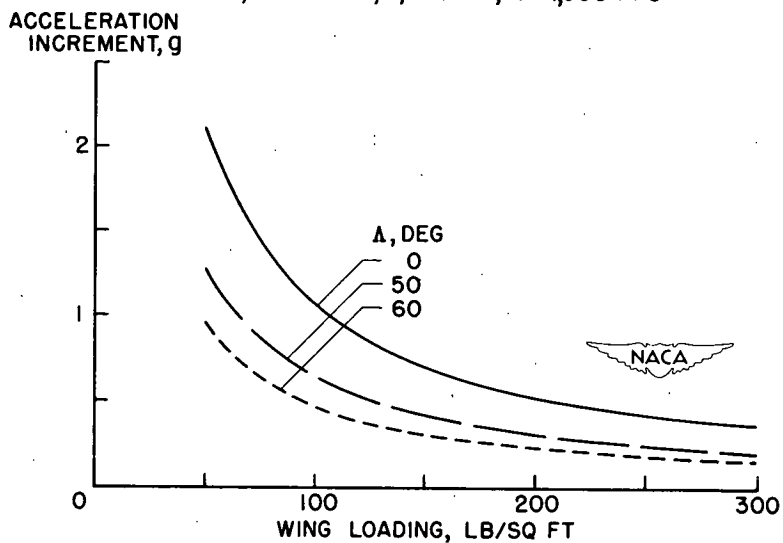
Figure 5

AIRPLANE CHARACTERISTICS
UNSWEPT WINGS

$\dfrac{dc_m}{dc_L}$	$\mu_g=10$ W/S=40 LB/SQ FT		$\mu_g=80$ W/S=160 LB/SQ FT		$\mu_g=200$ W/S=292 LB/SQ FT	
	P, SEC	$T_{1/2}$, SEC	P, SEC	$T_{1/2}$, SEC	P, SEC	$T_{1/2}$, SEC
-0.03	—	0.13	5.0	0.51	5.6	0.92
-.05	—	.12	3.7	.50	4.0	.91
-.08	—	.12	2.8	.49	3.1	.90

Figure 6

EFFECT OF STABILITY ON GUST ACCELERATIONS

dc_m/dc_L

—— o - 0.03
- - - - □ -.05
- · - △ -.08

$\left(\overline{\Delta n^2}\right)^{1/2}$

.32
.24
.16
.08

0   80   160   240   320

W/S, LB/SQ FT
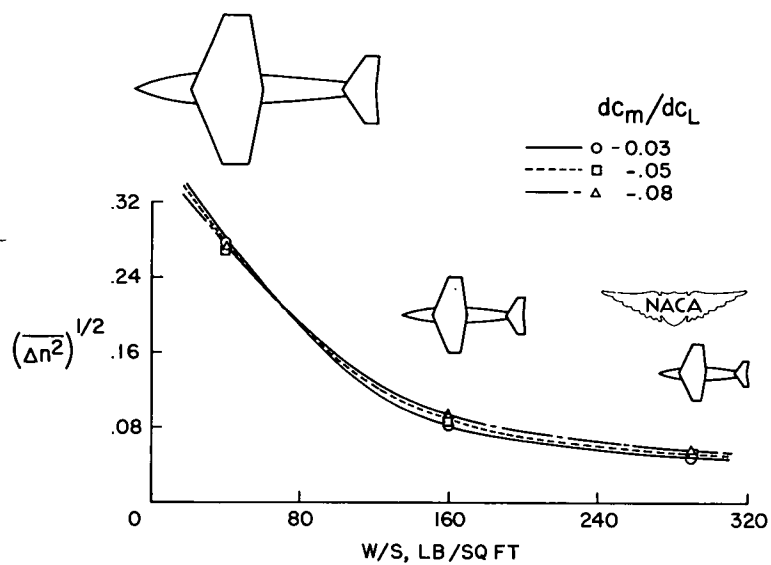
Figure 7

AIRPLANE CHARACTERISTICS
SWEPT WINGS

μ_g	$\Lambda = 0°$		$\Lambda = 45°$		$\Lambda = 60°$	
	$T_{1/2}$, SEC	W/S, LB/SQ FT	$T_{1/2}$, SEC	W/S, LB/SQ FT	$T_{1/2}$, SEC	W/S, LB/SQ FT
10	0.12	40	0.13	37	0.13	32
80	.48	160	.50	147	.54	127
200	.88	294	.92	270	.99	234
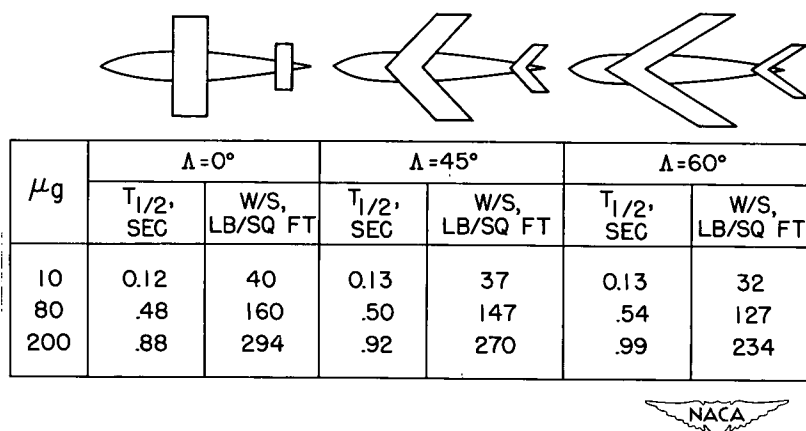
Figure 8

EFFECT OF PITCH ON GUST ACCELERATIONS FOR
SWEPT-WING AIRPLANES

$$\left(\overline{\Delta n^2}\right)^{1/2}$$

Λ, DEG

—o— 0

—□— 45

—△— 60

$$\dfrac{\left(\dfrac{dC_L}{d\alpha}\right)_{\Lambda=60°}}{\left(\dfrac{dC_L}{d\alpha}\right)_{\Lambda=0°}}$$

W/S, LB/SQ FT
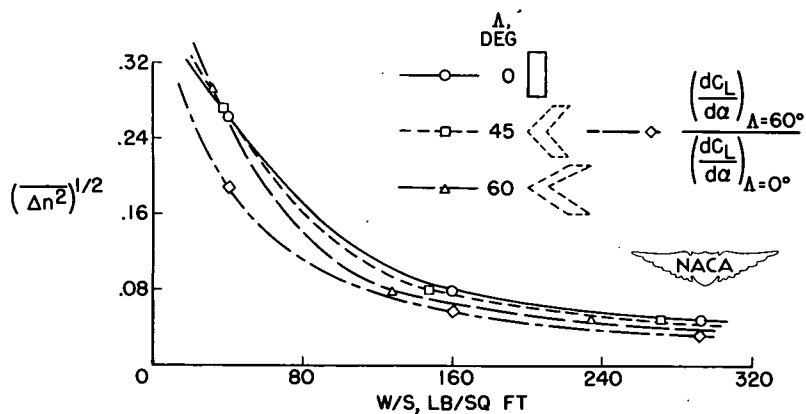
Figure 9

MODEL CHARACTERISTICS

WING
$A=6$
$\Lambda=45°$
$\bar{c}=9.85$ IN.

TAIL
$A=4$
$\Lambda=45°$
$\bar{c}=6.05$ IN.

57.9"

P, SEC

$C_{1/10}$, CYCLES

M

M
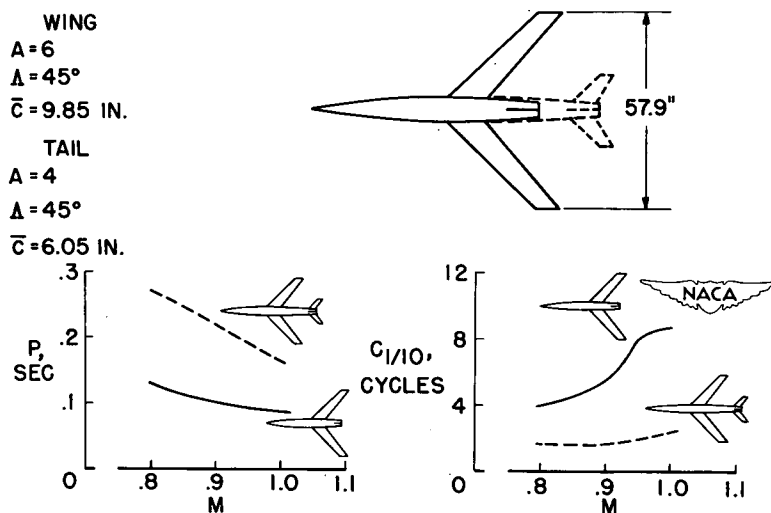
Figure 10

CALCULATED AND MEASURED POWER SPECTRUMS

M≈0.81
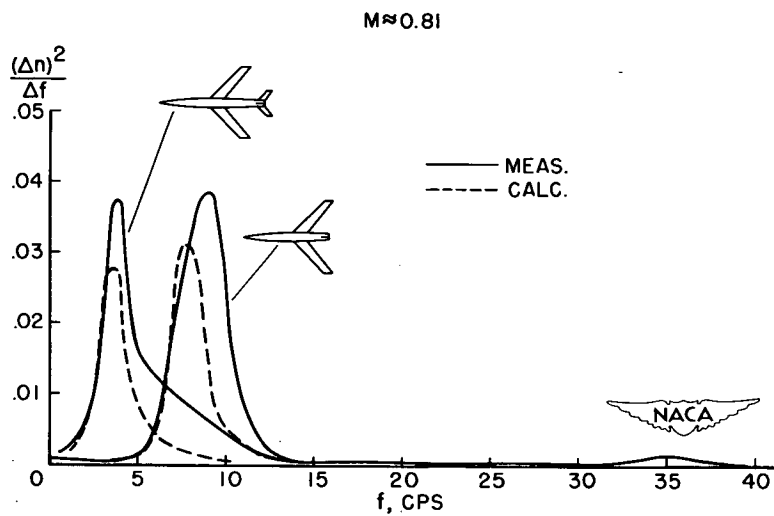
Figure 11

COMPARISON OF MEASURED AND CALCULATED RESULTS
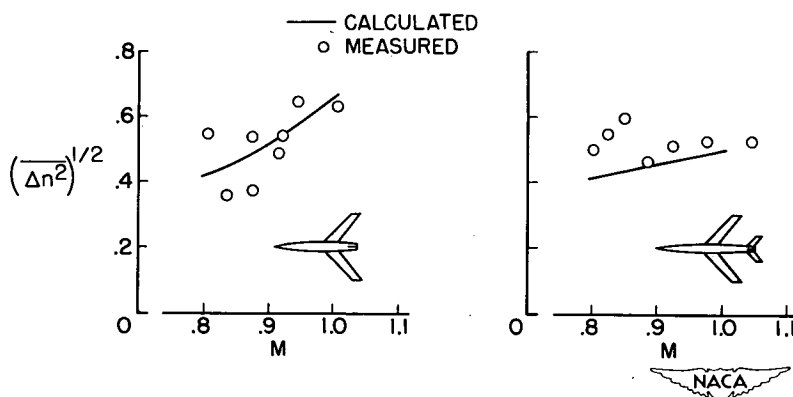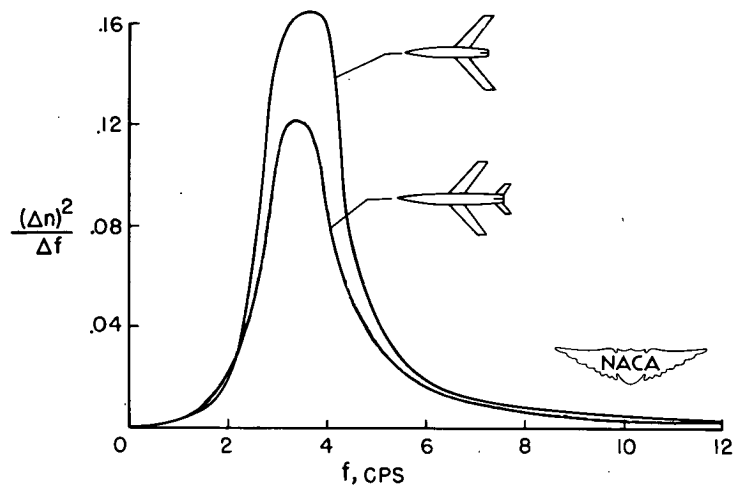

Figure 12

EFFECT OF CONFIGURATION

Figure 13(a)

EFFECT OF ARTIFICIAL DAMPING

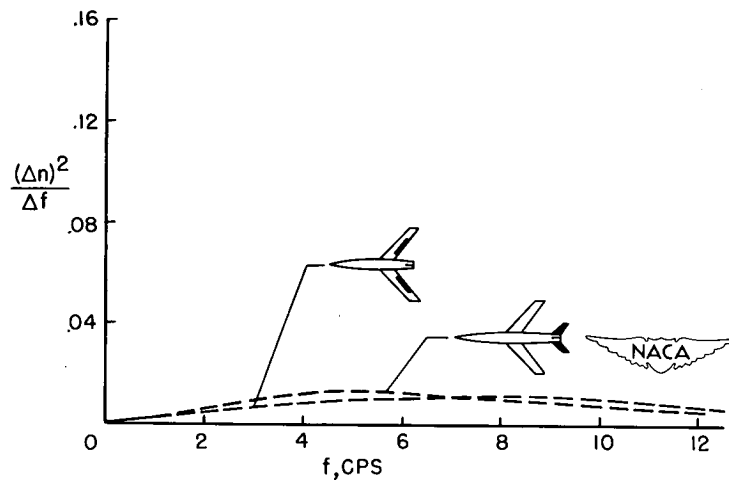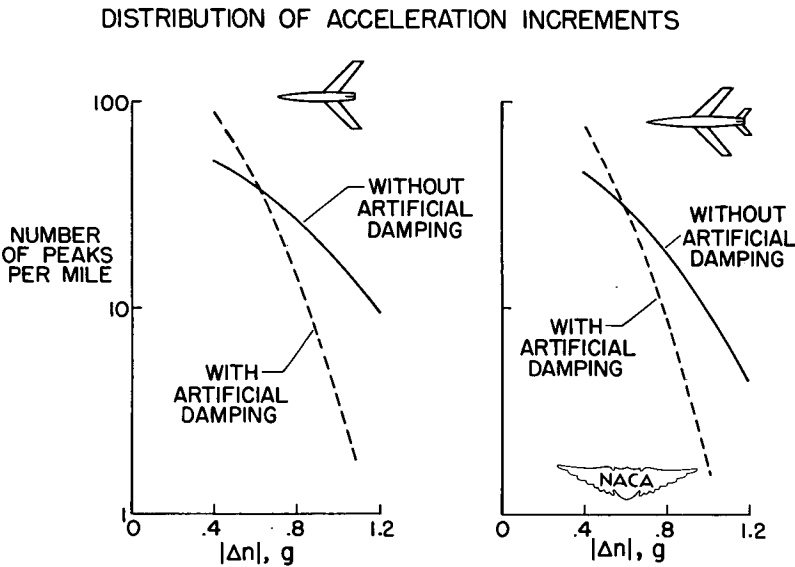

Figure 13(b)

DISTRIBUTION OF ACCELERATION INCREMENTS

NUMBER
OF PEAKS
PER MILE

100

WITHOUT
ARTIFICIAL
DAMPING

WITHOUT
ARTIFICIAL
DAMPING

10

WITH
ARTIFICIAL
DAMPING

WITH
ARTIFICIAL
DAMPING

1

NACA

0   .4   .8   1.2
|Δn|, g

0   .4   .8   1.2
|Δn|, g

Figure 14